MISSION DESIGN FOR A WRIGHT BROTHERS' CENTENNIAL MARS AIRPLANE MISSION IN 2003*

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Plans are underway at NASA and JPL to explore Mars on a global scale. Landers, penetrators, orbiters, and sample return are all included in the current roadmap. Airborne exploration can also be added, to provide broad and close-up views of inaccessible regions, such as the poles and canyons. Discovery proposals were submitted to NASA in 1998 detailing missions carrying planes or gliders. This paper is a summary of the mission design performed for sending gliders to Mars in 2003 as secondary payloads, on an Ariane GEO mission whose primary payloads would be communication satellites. In this analysis, a unique opportunity was uncovered whereby the gliders would arrive at Mars on December 17, 2003, or exactly 100 years after the Wright Brothers' first recognized flight at Kitty Hawk, NC. The launch process used to transfer the carrier spacecraft from the geosynchronous transfer orbit (GTO) to Earth escape will be described, as well as the development of a 3 to 5 month launch period, and the site accessibility regions at Mars, based on lighting, entry, and telemetry constraints. Finally, these results are applied to potential future Mars launch opportunities through 2009.

INTRODUCTION

Missions involving flights at Mars have been contemplated since the late 1960's. In these times of low cost exploration, it may be time to reconsider this option. Why planes? The reason is that they have the right characteristics for a low cost mission. For a science payload of as low as 1 kg, which could consist of several remote sensors as well as imaging, the plane can be very light weight, say 20-40 kg. Also, they can be designed and then tested in the upper atmosphere of the Earth. Once a plane is deployed, the Martian atmosphere can provide maneuverability, lift, temperature control, and stability. Also helpful is that light weight glider and plane technology is an active field, pursued by Aerovironment for example. At Mars, high-resolution imaging and spectral measurements can be made of the stratigraphic layers on slopes and canyon walls in close proximity for tens of kilometers. For the mission design performed here, flight time would be limited to 15-20 minutes during which time data would have to be returned to the carrier, and from there transmitted back to Earth.

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This mission poses some challenging mission design questions that must be explored. First, is there a low-cost launch mode which can be used for such a small payload? Then, what Earth-Mars trajectory will minimize the carrier propellant requirements, and deliver the gliders or planes to the desired location on Mars, and with acceptable lighting conditions? Finally, what would be the best mode to return the science data: use the carrier as a relay, or depend on an in-place Mars orbiter to relay the data? These questions are addressed here for the launch year of 2003.

A LOW COST LAUNCH MODE

Obviously, the way to minimize launch cost is to ride as a secondary payload on another carrier. A specific launch mode has been developed by this author[1] which uses the Ariane Structure for Auxiliary Payloads (ASAP)[2] offered by Ariane 4 or 5, which delivers communication satellites to GEO. A launch from French Guiana, near the equator, would place the spacecraft into a geosynchronous transfer orbit (GTO) after the primary payload is dropped off at GEO (35900 km altitude). Compared with LEO, this higher energy orbit would, under ideal conditions, require the carrier to apply only 1200 m/s to escape to Mars in 2003, instead of 3500 m/s. However, conditions are not ideal, so a rather circuitous route must be taken.

First, the rules of the game for secondary payloads specify that no conditions may be placed on the launch date or launch hour, except by the primary payloads. This implies that the highly eccentric GTO orbit will have a significant angular orientation of its axis in space, and will be near equatorial, for an Ariane launch. To escape using a single burn, not only would a hefty propellant penalty have to be paid for a plane change to get to a high escape latitude, if necessary, but also for a non-perigee burn if the GTO major axis is not oriented properly.

The multi-burn process proposed here for efficiently escaping from a generally oriented GTO orbit is depicted in Figure 1, and is referred to as Moon-Earth gravity assist, or MEGA. Three major burns are required to effect a near-minimum energy transfer to the required Earth escape vector for transfer to Mars.

Our starting point is the GTO, numbered (1) in Figure 1. The first maneuver (2) of about 750 m/s is made at the GTO perigee, and in-plane, to inject the spacecraft onto a large ellipse reaching out beyond the Moon, to a distance of 2 to 4 times the distance of the Moon. At apogee, about 10 to 30 days later, a second maneuver (3) of about 200 m/s will be performed to aim to a close lunar flyby (4), as required, to return directly to Earth with an altitude of 300 km (and proper inclination to achieve the escape latitude), where the third burn (5) of about 450 m/s provides the required escape energy to get to Mars. User interaction is required to satisfy the many boundary conditions imposed by this problem, where solar perturbations on the high ellipse must be taken into account. Note that varying the date, inclination, and altitude for the lunar flyby provides the degrees of freedom necessary to satisfy a desired Earth escape direction. The size of the third burn is determined by the escape energy required to get to Mars. The total propulsive cost for 2003, allowing 200 m/s for navigation and control, will be about 1600 m/s.

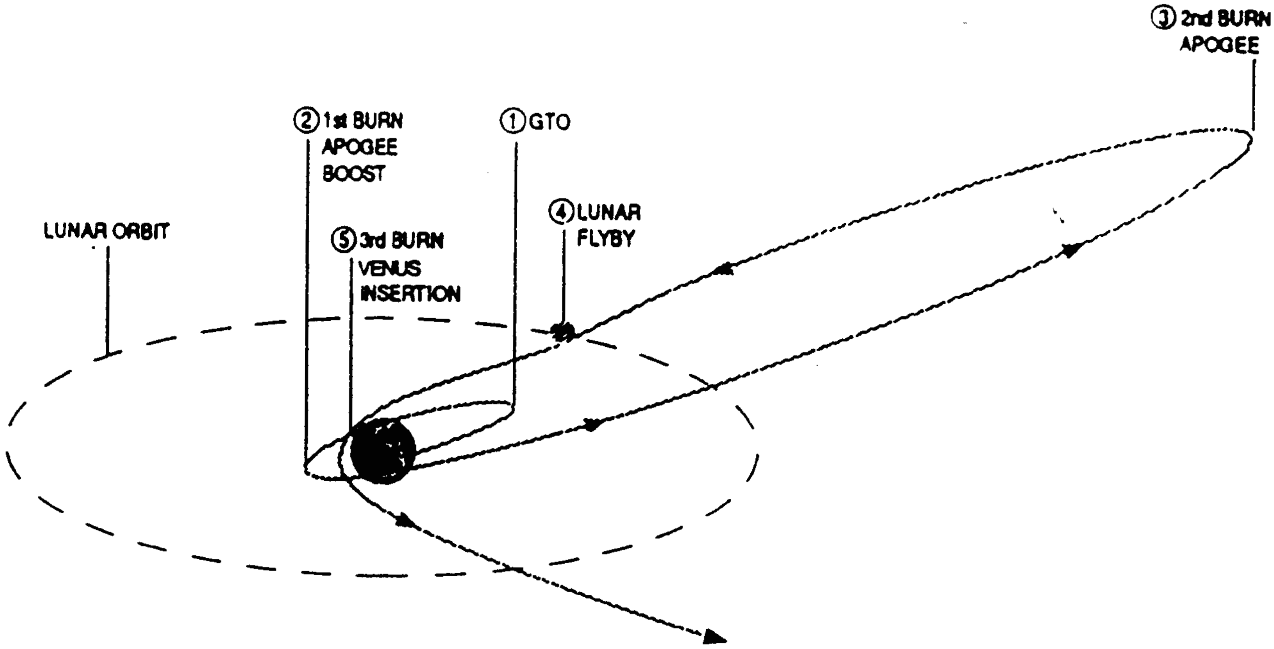

Fig. 1 GTO to Mars Using the 3-Burn Moon-Earth Gravity Assist

THE 3-BURN MEGA TRANSFER TO MARS

The design of the 3-burn trajectory begins with the third or escape maneuver. To start with, the position of the moon (and from this the flyby date) is related to the longitude of the desired Earth escape direction. For this mission, which requires minimum escape energy from Earth, the optimum launch opportunity occurs in mid-2003. Specifying the launch date and Mars arrival date, a conic approximation to the heliocentric transfer can determine the three dimensional outbound escape velocity vector from Earth, and the resulting inbound vector at Mars. That is, given two position vectors, from the sun to each planet, and the transfer time between them, a conic solution is given by Lambert's theorem.[2] (Figure 2 shows a typical Earth-Mars transfer.)

The specific components of these vectors, and other trajectory parameters, plotted as contours on launch-arrival date axes are available for reference.[3] The parameter of interest here is the right ascension of the launch escape vector, and this is given on the contour plot of Figure 3. The lighter contours are values of escape energy, C_3, which is the square of escape velocity.

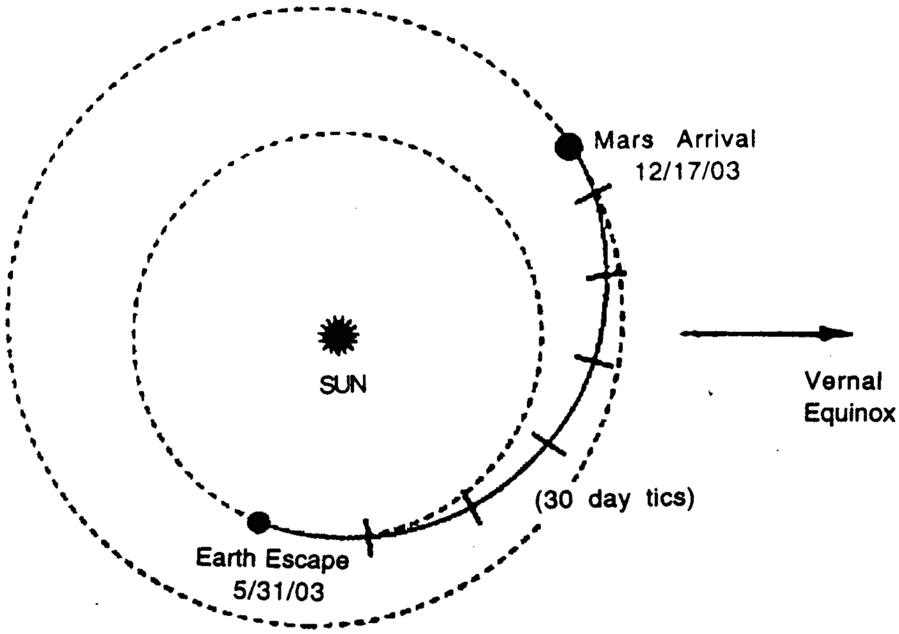

Fig. 2 Earth to Mars Heliocentric Transfer

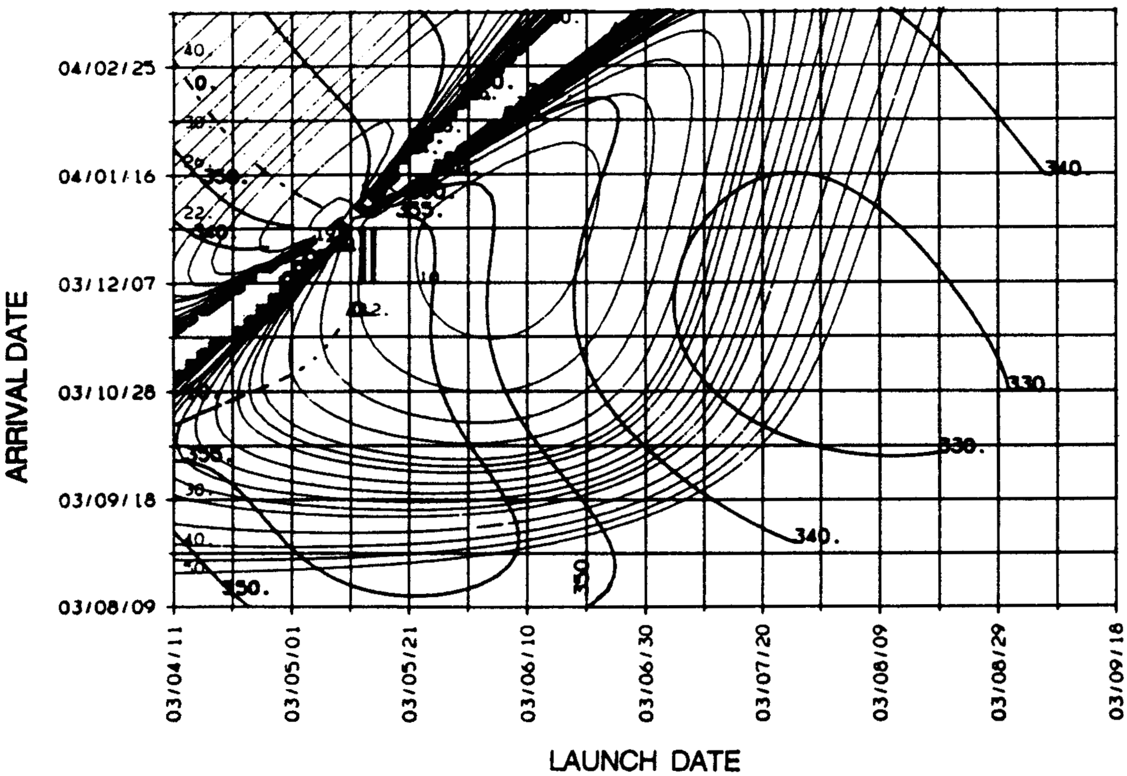

Fig. 3 Contour Plot for the Right Ascension of Earth Escape

Choosing the C_3 contour of 10 km^2/sec^2, it is seen that the right ascension does not change more than about 15°, from 340° to 355°, over the 30 day period, from May 24 to June 23. During this 30-day launch (or escape) period, there will be a particular day when the Moon is in the proper location for the flyby relative to this right ascension. As seen in the two plots of Figure 4, which are to scale, the angle in both plots between escape and the lunar flyby position is about 45°, so that the Moon's longitude should be about 35°. Accuracy is not too important here since the Moon moves about 13° per day, and we only want the day for starters. The lunar date, which gives a longitude close to 35° within our 30-day launch period, is May 28, 2003. The escape date will then be about 3 days later, at the third burn, or about May 31, 2003.

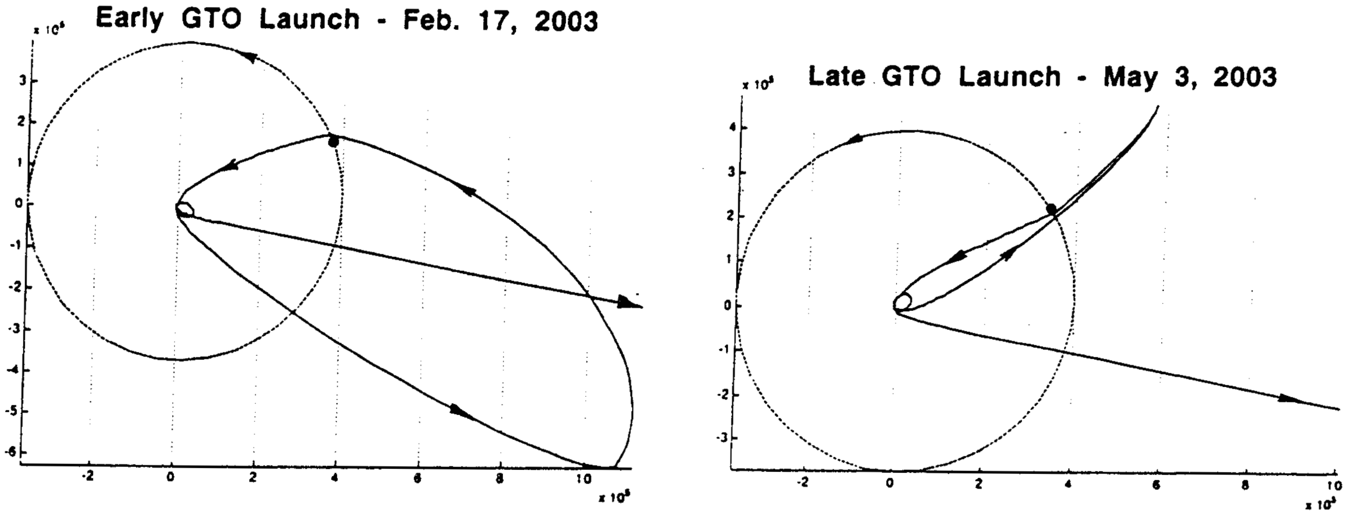

Fig. 4 Polar View of 3-Burn MEGA Trajectories for Mars 2003

The contour of Figure 3 also bounds the Mars arrival date, for the launch date of May 31, and these dates are between November 17, 2003 and January 11, 2004. But the best choice would be that date which produces the minimum C_3, which is December 17, 2003. The value here is about 9 km^2/sec^2. Presenting this data at a team meeting, it was soon realized that this is the date of the 100th anniversary of the Wright Brothers' historic first flight.

THE ARIANE LAUNCH PERIOD

Serious consideration of launching planetary spacecraft on the Ariane GTO missions as secondary payloads was given after this was suggested by J. Blamont of CNES.[4] The MEGA approach was developed about a year later. The launch of the Ariane is from Kourou in French Guiana, located at 5°N latitude and 53°W longitude. Launch is just south of east into a nearly equatorial orbit. Figure 5 gives more specific orbit parameters for the GTO launches and the Ariane 4.[5] The Ariane 5 parameters will be similar. The time from launch to the GTO injection maneuver is about 30 minutes.

To proceed to the MEGA calculations, the GTO orbit must be known in the inertial frame, preferably relative to the Ecliptic 2000 system. Since the orbital parameters given in Figure 5 are in the Earth rotating frame, a perigee passage date, or the equivalent, is needed. Ariane launch statistics indicates that this time is near midnight UT, the rationale being that arrival at apogee where the communication satellites are released will be in full sun to perform the terminal maneuvers into their GEO orbits. This noontime orientation is reflected in Figure 4, where the vernal equinox points to the right. That is, on February 17, the Earth is at about 150°, and on May 3, it is at about 225°. The sun, then, points in the opposite direction, or at 330° and 45°, respectively, which are the directions of the high ellipses due to the first burn.

Ariane standard Geostationary Transfer Orbit (GTO) parameters.

The orbit is defined in terms of osculating parameters at injection as follows

inclination	i	= 7.0°
altitude of perigee	Zp	= 200 km
altitude of apogee	Za	= 35 975 km
argument of perigee	ω	= 178°

. Injection is defined as the end of third stage thrust-decay
. Za is equivalent to true altitude 35 786 km at first apogee
. The longitude of the first descending node Ω as defined in figure 2.1, depends upon the particular Ariane 4 configuration being used and normally lies around 11° (West).

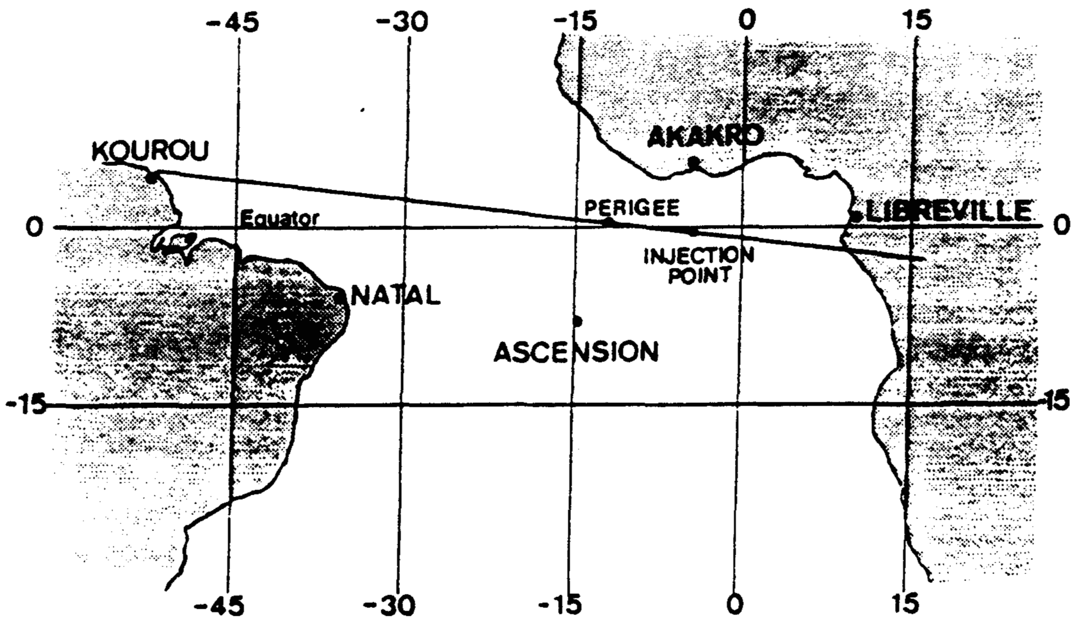

Figure 2.1 - Typical ground track

Fig. 5 Ariane Standard Geostationary Transfer (GTO) Parameters

In Figure 4, for the February 17 launch, there are 100 days before the lunar flyby on May 28. For the first 40 days after GTO launch and release, the spacecraft is assumed to remain in GTO, or enter intermediate phasing orbits. Then, the maneuver into the high ellipse shown is applied on March 29 leaving 60 days to get to the lunar flyby. On the other hand, the May 3 launch has 25 days to the lunar flyby, and the high ellipse burn is executed soon after GTO launch. Note that, for this case, the apogee maneuver of almost 200 m/s results in a retrograde trajectory to the Moon. This apogee maneuver is the major contributor to the variation in the value of the 3-burn velocity requirement. A plot of this requirement is given in Figure 6.

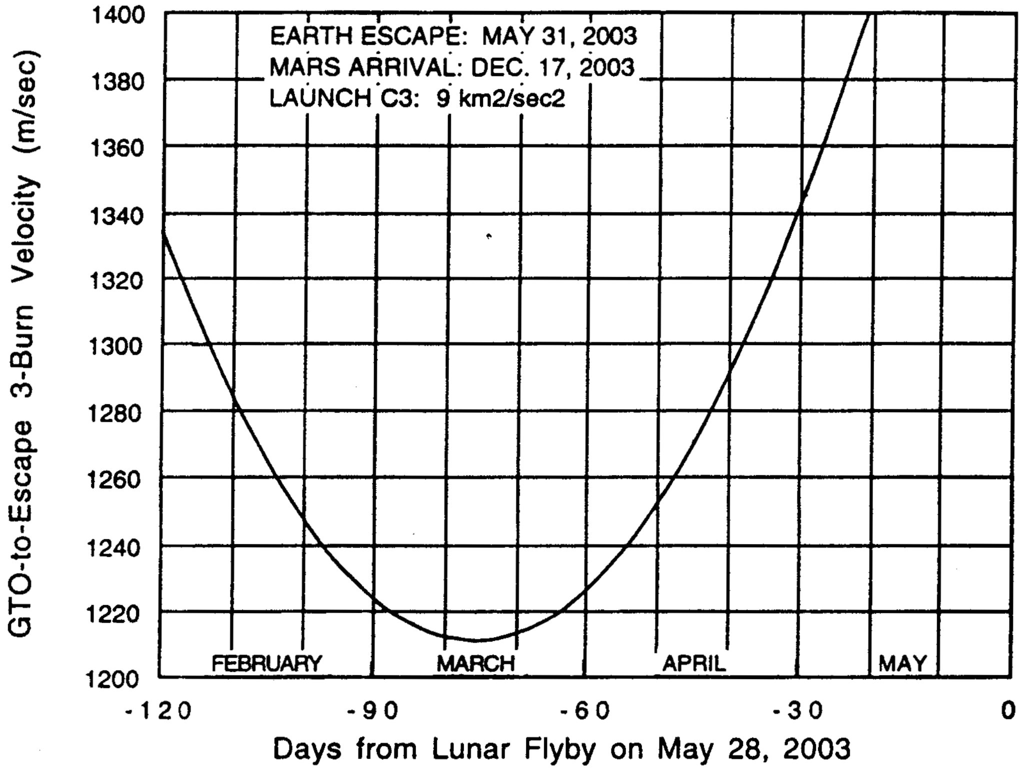

Fig. 6 The 3-Burn Velocity for High Ellipse Noon Orientation

Assuming a noontime GTO apogee arrival for the Ariane launch, the 3-burn MEGA process can handle about 100° variation in the GTO major axis orientation. This variation can be due to a varying launch date for a fixed launch hour, or a 1° per day change, or it can be due to a varying launch hour for a fixed launch day or 15° per hour, or both. Extensions to the 3-burn process can be made to increase the launch day-launch hour window by using the 5- or 7-burn mode[6]. The 5-burn for the Mars 2003 trajectory is shown in Figure 7. Basically, an extra 50 to 60 day high ellipse is inserted to rotate the existing major axis this many degrees, so that the new orientation will represent the orientation of a high ellipse of a launch two months later. In this manner, the length of the launch period can be unlimited.

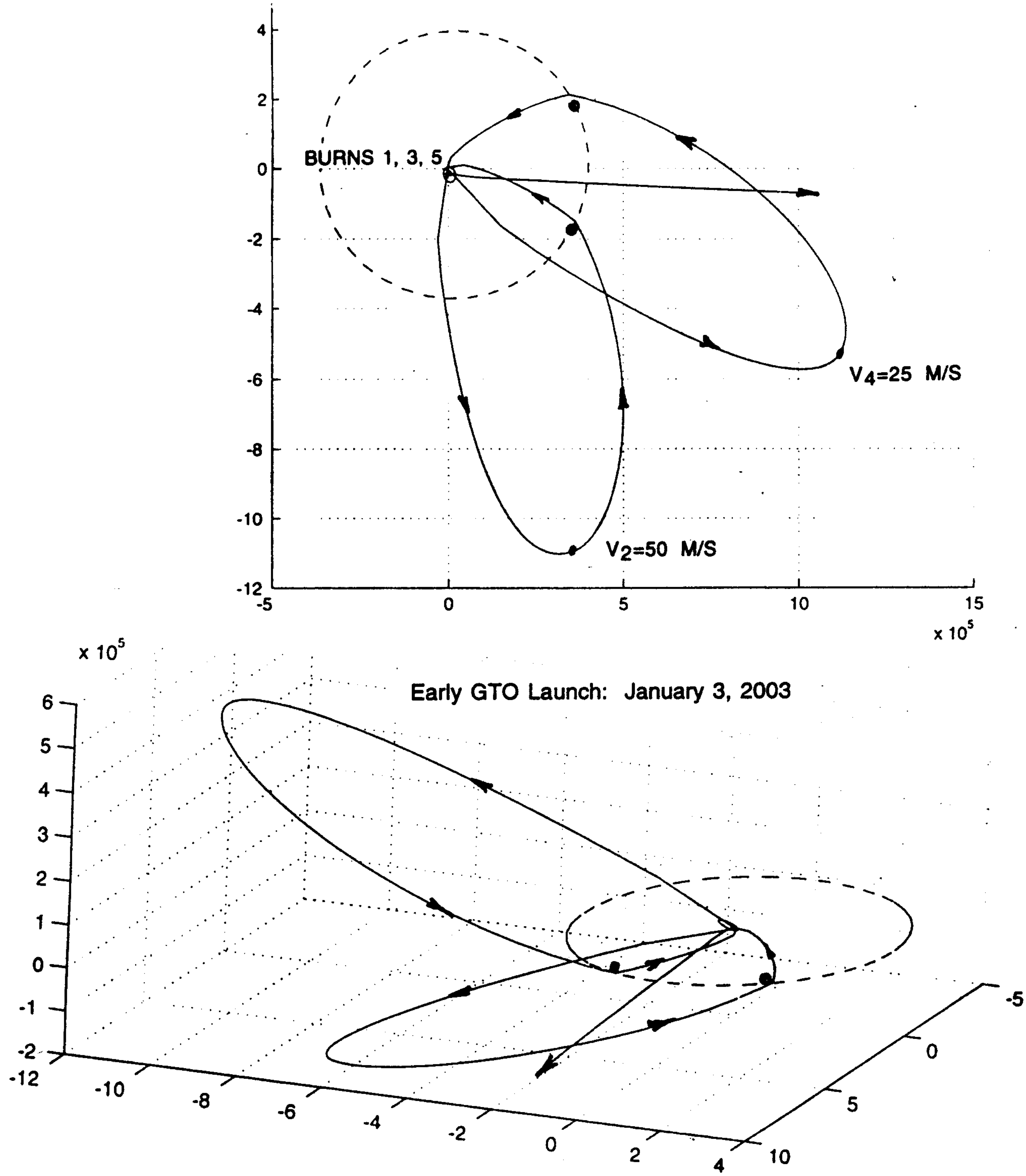

Fig. 7 The 5-Burn Trajectory Views for Mars 2003

MARS APPROACH GEOMETRY AND SITE SELECTION

The heliocentric Lambert conic solution to the Earth-Mars transfer shown in Figure 4 also provides the sun-relative velocities at Earth departure and Mars arrival. Subtracting off the Earth's velocity, vectorially, gives the Earth relative departure escape velocity. (Earth and Mars are only considered massless points in this calculation.) A similar subtraction at Mars gives the Mars approach velocity vector, which will move along an approach hyperbola, which is produced once Mars gravity takes effect.

If the aiming at Mars were such that the periapsis altitude is above the atmosphere, the spacecraft would fly by Mars and depart on the outgoing asymptote. This, then, would be a Mars gravity assist, assuming that it is headed somewhere. However, given a fixed flight path (or entry) angle at some low altitude at Mars, the shape of the approach hyperbola would be completely determined, regardless of whether it would fly around the equator or over one of the poles. Figure 8 presents the path of several of these hyperbolas for a fixed entry angle of -12° at an entry altitude of 125 km. Since all of these hyperbolas are the same, except for inclination, the locus of the possible entry points will make a small circle on the backside of Mars, and the planes of these hyperbolas will contain the incoming approach velocity vector. The entry locus for -20° entry angle is also shown.

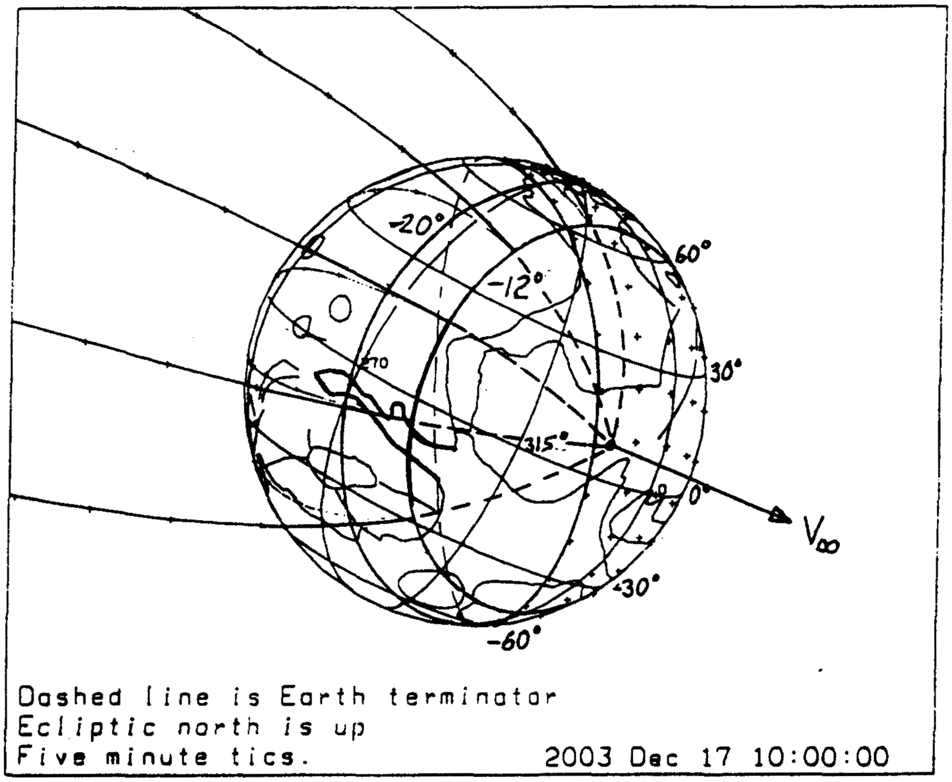

Fig. 8 Entry Loci for the Mars 2003 Mission

This geometry for the hyperbolas is inertially fixed, and does not change appreciably during a daily Mars rotation. Also, only small corrections are needed enroute to Mars to change the arrival time by hours, so that for any given approach inclination, any given surface longitude is available. The entry latitude will have limits for a given entry angle, ranging from a maximum for approaches over the north pole of Mars, to a minimum for a flight over the south pole. These limits are shown in Figure 9, together with curves of entry azimuth and sun elevation, as a function of entry latitude.

Of particular interest as an entry site, is Valles Marinaris, a deep canyon at -10° latitude, and extending from 270° to 315° longitude. Figure 8, with Valles Marinaris in bold outline, shows that arrival on December 17, 2003; 10 hr GMT, would place the Mars planes at this location with the entry angles shown, in daylight, and in the Mars afternoon. Figure 9 below shows that the entry azimuth will lie in the 72° to 78° range, and that the sun elevation will lie in the 48° to 66° range, as the entry angle goes from -12° to -20°. plots of other parameters, such as the sun azimuth, are available but not shown here.

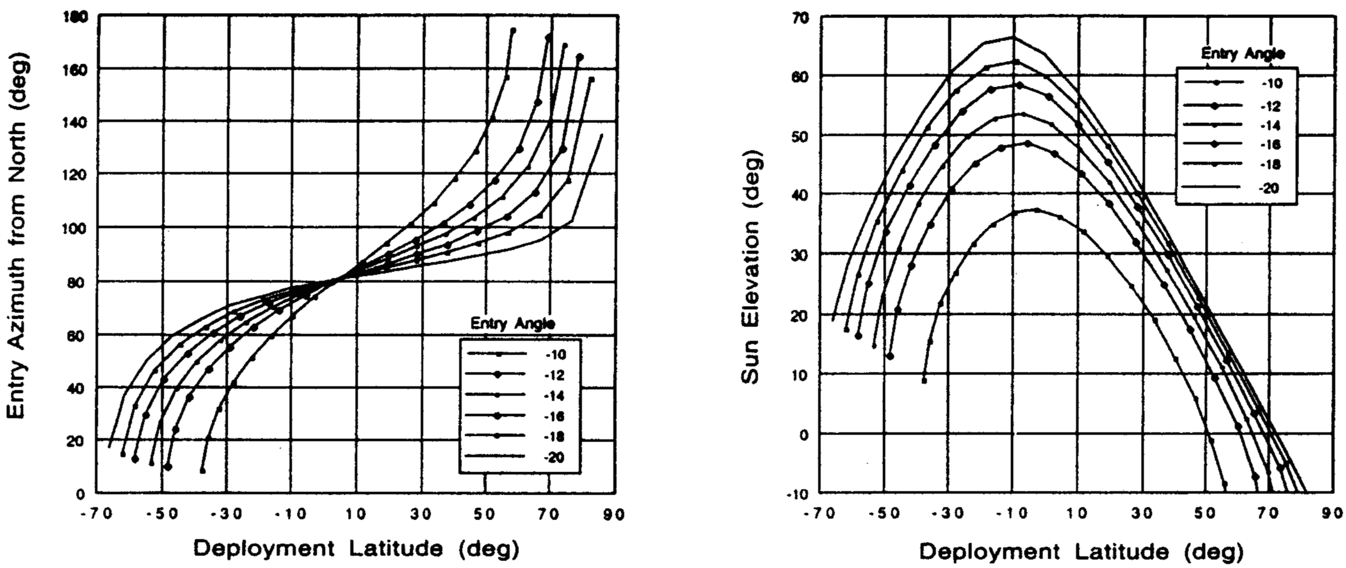

Fig. 9 Entry Azimuth and Sun Elevation for the Mars 2003 Mission

CARRIER SPACECRAFT DATA RELAY GEOMETRY

Two methods were considered for returning the science data to Earth once it has been received by instruments aboard the airplane: (1) relay the data to a Mars orbiter, which would store and later transmit the data to Earth, or (2) have the carrier fly over the planes and receive the data. The data would then be stored and transmited back to Earth as it departs Mars. Using the carrier has the advantage that it is only necessary to raise the periapsis altitude, and time the overflight so that it is in phase with the planes transmissions. The orbiter, on the other hand, would have to be launched with the carrier on the Ariane, which has been considered, or otherwise be available and synchronized with this mission event.

The carrier delta-v requirements are modest, less than 10 m/s at five days out. A typical result showing slant range and elevation of the carrier relative to the planes is given in Figure 10. Here, the entry angle is -20° and the periapsis altitude is 900 km. Included is a downrange angle from entry to parashute deployment where it is assumed the relay of data can begin. A slant range limit of 5000 km was imposed by the data rate required. Assuming an elevation limit of 20°, the duration of the relay is about 20 minutes, which is the assumed duration of a glider mission based on other factors. A higher altitude resulting in a longer relay duration is possible, but has not been studied in detail.

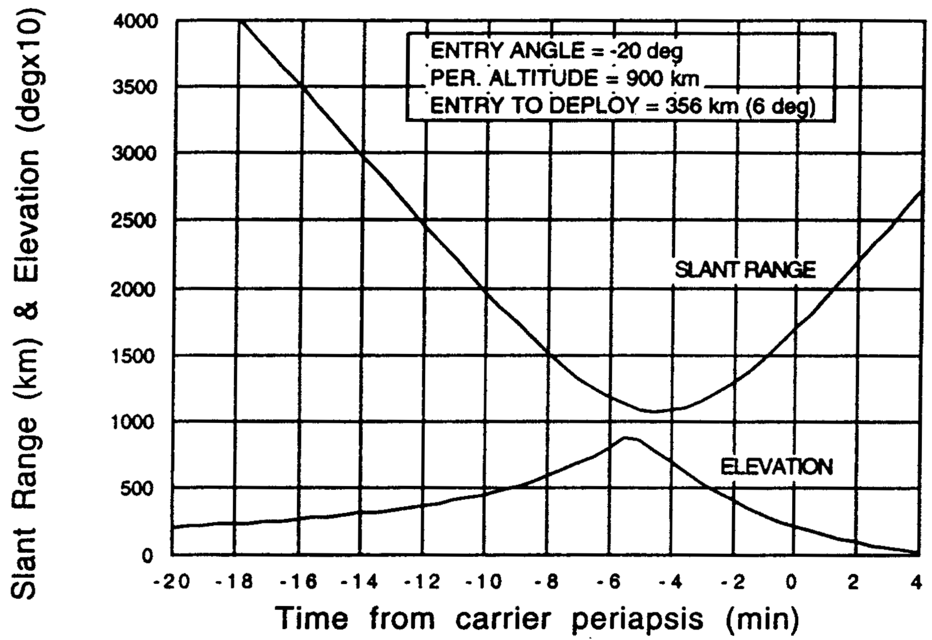

Fig. 10 Slant Range and Elevation of the Carrier During Data Relay

FUTURE MARS AIRPLANE MISSION OPPORTUNITIES

The first in-depth studies of sending planes or gliders to Mars have taken place only in the past 2 years, and only for the 2003 launch opportunity. This will likely be missed, but consideration may be given for other years. With respect to lauches as secondary payloads in future years, the propellant requirements will be higher. For example, the year 2003 can be compared with future launch opportunities, and this comparison is given in the table below.

Presumably, at some time in these later years, experience will have been gained with Ariane launches to Mars, and telecom orbiters may be in place to assist in gathering science data and relaying it to Earth. With the aggressive Mars exploration program now planned for Mars, it is very possible that planes may be flying in the Martian sky sometime in the future. Only time will tell.

TABLE - TO BE ADDED

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1. Penzo, P. A., "Planetary Missions from GTO Using Earth and Moon Gravity Assists", Paper AAS 98-4393, presented at the AIAA/AAS Astrodynamics Specialist Conference, Boston, MA, 10-12 August 1998.

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4. Blamont, J., "Using Large Launchers for Small Satellites," JPL Publication 96-26, Introductory Lecture to the 10th Annual AIAA/Utah State University Conference on Small Satellites, 16 September 1996.

5. *Ariane Structure for Auxiliary Payload (ASAP) 5 Users Manual*, published by Arianespace, May 1997.

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Table 1. Carrier Missions for Mars 2003-2011

EARTH ESCAPE DATE	C_3 (km^2/s^2) ENERGY (TYPE)	GTO LAUNCH PERIOD	MEGA* ΔV (m/s)	MARS ARRIVAL DATE	MARS# VEL./LAT (km/s,deg)	LATITUDE RANGE** (deg)
2003 MAY 31	9(1)	FEB 2-MAY 3	1600	03DEC 17	2.75/10	87/-66
2005 AUG 4	16(1)	APR 10-JUL3	1900	06FEB 22	3.17/11	89/-69
2005 SEP 2	16(2)	MAY 3-AUG 5	1900	06OCT 2	3.44/-27	55/-71
2007 SEP 13	14(2)	MAY 16-AUG 15	1800	08AUG 19	2.51/-24	50/-50
2009 OCT 18	11(2)	JUN 17-SEP 20	1700	10AUG 27	2.49/0	74/-74
2011 OCT 30	10(2)	JUL2-OCT 4	1600	12SEP 8	2.73/19	85/-57

*Navigational Corrections(150 m/s), Mars Maneuvers(50 m/s) Added
#Mars Approach V-Infinity Velocity Magnitude and Latitude
**Applies to -20 deg Entry Angle plus 6 deg from Entry to Deployment

REFERENCES

1. Penzo, P. A., "Planetary Missions from GTO Using Earth and Moon Gravity Assists", Paper AAS 98-4393, presented at the AIAA/AAS Astrodynamics Specialist Conference, Boston, MA, 10-12 August 1998.

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